NASA Technical Memorandum 104493
AIAA-91-2176

A Review of Candidate Multilayer Insulation Systems for Potential Use on Wet-Launched LH_2 Tankage for the Space Exploration Initiative Lunar Missions

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Prepared for the
27th Joint Propulsion Conference
cosponsored by the AIAA, SAE, SAME, and ASEE
Sacramento, California, June 24-27, 1991

NASA

A REVIEW OF CANDIDATE MULTILAYER INSULATION SYSTEMS FOR POTENTIAL USE ON WET-LAUNCHED LH_2 TANKAGE FOR THE SPACE EXPLORATION INITIATIVE LUNAR MISSIONS

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E-6351

SUMMARY

The storage of cryogenic propellants such as liquid hydrogen (LH_2) and liquid oxygen (LO_2) for the future Space Exploration Initiative (SEI) will require lightweight, high performance thermal protection systems (TPS's). For the near-term lunar missions, the major weight element for most of the TPS's will be multilayer insulation (MLI) and/or the special structures/systems required to accommodate the MLI. Methods of applying MLI to LH_2 tankage to avoid condensation or freezing of condensible gases such as nitrogen or oxygen while in the atmosphere are discussed. Because relatively thick layers of MLI will be required for storage times of a month or more, the transient performance from ground-hold to space-hold of the systems will become important in optimizing the TPS's for many of the missions. The ground-hold performance of several candidate systems are given as well as a qualitative assessment of the transient performance effects.

INTRODUCTION

The storage of cryogenic propellants such as liquid hydrogen (LH_2) and liquid oxygen (LO_2) for the future Space Exploration Initiative (SEI) will require lightweight, high performance thermal protection systems (TPS's). The TPS's will be mission dependent, but most cryogenic tanks will require multilayer insulation (MLI) systems, low-conducting tank supports and propellant lines, low-conducting electrical and instrumentation leads, and selective thermal control surfaces. For the longer duration missions to Mars or for extended cryogenic storage in a space-based depot or on the lunar surface, the TPS's may also include vapor-cooled shields, para-to-ortho converters, refrigeration systems[1] and/or shadow shielding[2] or combinations thereof.

For the near-term lunar missions, the major weight element of the TPS's will be the MLI and/or special structures/systems to accommodate the MLI insulation systems. Most of the cryogenic tank sets in the recent studies on "Human Exploration of the Moon and Mars" require wet-launched tanks or tanks that are filled with cryogenics (i.e., LH_2 and/or LO_2) during pre-launch and launch operations within the Earth's atmosphere. Launching full tanks of LH_2 requires special accommodations and/or procedures to avoid condensing or freezing ambient air or nitrogen purge gas within the MLI while within the atmosphere. This can be accomplished by adding a sealed sublayer of insulation, such as foam, beneath the MLI or by purging the MLI with helium (He) or combinations of a dry N_2 purge and a helium purge.[3] Condensation of gases within the MLI can cause a considerable delay in achieving a good vacuum subsequent to launch and hence degrade the thermal performance of the MLI[4] due to the additional heat transferred by gaseous conduction.

This report reviews the general requirements of MLI for the lunar missions and the current status of MLI as applied to flightweight cryogenic tankage. Several candidate MLI systems for use on wet-launched LH_2 tanks are then presented and, finally, a general MLI technology enhancement program to meet the needs of the SEI lunar missions is discussed.

GENERAL MLI REQUIREMENTS FOR THE SEI MISSIONS

An indication of the capability of MLI to achieve the thermal performance levels required for the SEI missions is shown in Fig. 1. Heat transfer rates of MLI insulated LH_2 tanks and/or calorimeters (at 37 °R), with a hot side boundary temperature of 530 °R, are shown for various numbers of radiation shields in the MLI or MLI thicknesses. The layer density of the data presented range from roughly 30 to 60 radiation shield per inch.* Some representative mission requirements for in-space LH_2 tanks from early SEI mission studies are also shown on the right of Fig. 1. It is apparent that the existing data base for MLI is for insulation thicknesses considerably thinner than those needed for future SEI missions. Although lower hot side boundary temperatures will be expected for many of the SEI applications (which will reduce the heat transfer rate and thereby give some performance improvement), it is apparent that the MLI data base must be extended to much thicker systems, especially for tank-applied systems. Also evident from Fig. 1 is the large disparity between calorimeter data and tank-applied data. The circle symbols, for example, are for an identical 34-layer MLI system and give a direct comparison between calorimeter and tank applied data. The differences are largely due to performance degradations caused by seams and penetrations in the tank applied systems. Hence, SEI applications will require considerable design discipline and attention to detail, in fabrication and assembly, especially for smaller tanks.

*Details of the systems tested, as noted by the various symbols, are given in Appendix A.

Even with great care, however, it is apparent that MLI alone cannot afford the necessary protection for the longer missions (>several months) and other thermal control means such as vapor-cooled shields, shadow shielding and/or refrigeration systems will be required.

In order to assess the MLI thickness levels required for the near-term lunar missions, typical boil-off rates and MLI insulation system weights were calculated for an insulated LH_2 tank in low Earth orbit (LEO). The results for an assumed outer insulation temperature of 450 °R are shown in Fig. 2. These data are linearly extrapolated from a nominal 0.1 Btu/hr ft^2 performance level* assumed for 1.5 in. of MLI (~90 layers) which should be achievable for larger sized tanks. The ordinate on Fig. 2 gives both the boil-off loss expected, and the estimated insulation weight, on a unit area basis. The MLI insulation system is made up of 0.5 in. blankets with each blanket consisting of two outer reinforced cover sheets (radiation shields) with 28 layers of MLI in between. The change in slope at 1.5 in. is caused by adding sufficient foam beneath the purge bag to insure that at least a minimum of 1.5 in. of helium purged insulation is available for ground-hold thermal protection which should give heating rates on the order of 100 Btu/hr ft^2 or less.

Although the TPS weight optimization of an MLI-insulated tank involves many other factors (e.g., tank volume changes, propellant usage schedule, etc.), a first order estimate of an optimized system is where the boil-off and insulation weight curves cross. Using this as an approximate

*The assumed performance level is an engineering judgement based on extrapolating the tank applied data in Fig. 1 and adjustments for lower temperatures.

indicator of the MLI thicknesses required, it is apparent that MLI thicknesses of 1.5 in. or more will be needed for the current storage periods envisioned for the lunar mission tank sets (45 days minimum and up to 180 days on the lunar surface).

In addition to MLI thicknesses that exceed our existing data base, especially for tank applied systems (Fig. 1), the expected hot side surface temperatures of the MLI will also vary widely from that for which the existing data base was obtained (primarily 530 °R). These temperatures will range from approximately 400 to 450 °R in low Earth orbit to lower values during transit and temperature swings from nominally <200 °R to >600 °R on the lunar surface. (Transit temperatures from Earth to Mars will be 300 °R maximum, with no preferential vehicle orientation to less than 100 °R with vehicle orientation.)

Finally, the current baseline for the lunar missions for the SEI uses the tank change-out concept which requires most of the tank sets to be wet-launched (i.e., launched full). The required MLI systems then must perform adequate thermal protection functions, both during ground-hold while within the atmosphere and in-space while within a vacuum. Conventional ground-based LH_2 storage tanks use a double-walled vacuum jacket to provide the vacuum environment required for the MLI (required to achieve the desired low heat flux performance). These relatively heavy vacuum jacket systems, however, are not practical for large flightweight tankage. Various lightweight systems have been proposed to provide ground-hold thermal protection for MLI insulated LH_2 tanks.[3,4] Two examples of these are given in Fig. 3. The purpose of these systems is to preclude any condensation and/or freezing of air or other condensibles while within the atmosphere and to provide various means of allowing the MLI to pump down to a vacuum environment as rapidly as possible, once orbital conditions are achieved. In the first concept, the MLI system is slowly purged with gaseous helium (GHe), and the MLI performance during ground-hold is essentially that predicted by pure conduction through the quiescent helium within the MLI thickness. The second concept uses a base insulation, such as a sealed foam, of sufficient thickness to raise the foam surface temperatures well above 160 °R to preclude condensing or freezing GN_2 or dry air. This system performs better during ground-hold (because the thermal conductivity of both the foam and N_2 purge gas are significantly lower than that of helium), but must maintain a near perfect seal at the foam interface to prevent condensate or solids from forming within the MLI system. The condensate or solids will severely degrade the on-orbit performance of the MLI due to gaseous conduction during the prolonged period required to achieve a good vacuum (i.e., 10^{-6} torr).

CURRENT STATUS OF TANK APPLIED MLI

To date, the only lightweight application of MLI to LH_2 propellant tankage was the Centaur D1-T which used 23 layers of MLI on the forward bulkhead and 3 layers of MLI on the tank sidewall. Ground-hold thermal protection was provided by a selective gaseous helium purge in the Centaur-stage section of the insulated shroud that enclosed both the Centaur-stage and its payload.

Another lightweight MLI system that was developed (but never flown) for a LH_2 tank was that for the Shuttle/Centaur G Prime vehicle shown in Fig. 4.[6] The insulation system for the tank sidewall consisted of two 0.75 in. thick helium purged open cell polyamide foam layers overlaid by three radiation shields or three layers of MLI

(Fig. 5(c)). The VDA referred to is vapor deposited aluminum. The inner shield served as the helium purge containment membrane while the outer two shields were perforated to rapidly evacuate once on-orbit. Results from calorimeter tests on the sidewall insulation system gave ground-hold heat transfer rates of approximately 100 Btu/hr ft^2 or less (also confirmed on full scale vehicle tests), whereas the on-orbit performance was 1 Btu/hr ft^2 or less (for a 480 °R outershield temperature). Modified versions of this system are expected to be flown on the Titan-Centaur in the near future.

Although the Centaur represents the only lightweight application of MLI on an operational LH_2/LO_2 propellant stage, a limited amount of research was also performed on tank-applied MLI (sponsored by the Lewis Research Center and the Marshall Space Flight Center) in the 1960's and early 1970's. These were represented by the solid symbols in Fig. 1 and are reported on in Refs. 7 to 10. One representative of a tank applied MLI system[7] is shown in Fig. 6. This system had 34 layers of MLI and was extensively tested in a vacuum at a 530 °R hot side boundary temperature as well as for simulated interplanetary missions where the tank was shaded from the sun by preferential vehicle orientation.[2,7] This tank has also been recently tested for a range of MLI insulation surface temperatures expected on the lunar surface.[11]

In more recent efforts, the Air Force Systems Command Astronautics Laboratory now has a contract underway (F04611-90-C-0131 "Long Term Cryogenic Storage Demonstration") where they plan to fabricate a TPS consisting of a 6-in. thick MLI system (with vapor-cooled shields) for a 10 ft diameter tank and perform both ground-hold thermal tests and in-space thermal tests. The tests are planned for 1993 and evolved from the work of Ref. 12.

None of the tank applied MLI systems were tested for the complete operational conditions expected for wet-launched cryogenic tanks, that is, ground-hold, rapid depressurization during launch, and the subsequent transient to an on-orbit performance. Some limited calorimetric data on wet-launched tank MLI concepts were acquired in the early 1970's.[4] Representative data for three 30-layer MLI systems from Ref. 4, which were tested on a 30 in. diameter cylindrical LH_2 calorimeter, are given in Fig. 7. The transient heat flux as a function of time for a typical ascent pressure profile (simulated Saturn V launch) is given for all three systems along with their steady state performance under hard vacuum conditions. The ground-hold thermal performance indicated by the data at 0.1 min is as expected, that is, the system with the thinnest fiberglass mat sublayer gave the highest heat flux and the system with the foam sublayer gave the lowest ground-hold heat flux. The steady state data at the extreme right of each figure was measured in separate tests and was essentially a function of the vacuum conditions (indicated in parenthesis) achieved during those tests.

The transient data from Fig. 7 indicates that on-orbit performance of the MLI can be achieved in nominally an hour and, in fact, the performance of the systems with the fiberglass mat substrate actually show heat transfer rates less than that measured in steady state tests. It is suspected that the MLI insulation systems were not yet in thermal equilibrium and that if the tests would have continued longer, the heat transfer rates would have eventually increased back up to their steady state performance levels. Although not shown, the MLI interstitial pressure lagged the external environmental (chamber) pressure during the transient pumpdown for all the systems tested. The resulting transient thermal performance is dependent on the initial MLI temperatures during the ground-hold phase, the pressure history of the MLI, and the subsequent temperature changes within the insulation system as it is evacuated to 10^{-6} torr or less. The MLI system with the foam does not dip below its steady state performance because of the relatively high temperature imposed on the MLI during ground-hold conditions and because of the heat stored in the foam. The thickness of foam used was not optimized for these tests and could be much thinner as pointed out by the author of Ref. 4. (Using thinner layers of foam as is currently being done in the MSFC efforts described in Ref. 13 should result in less stored heat and lower MLI temperatures.)

One of the original intents of the work reported in Ref. 4 was to use a gaseous N_2 purge in the MLI which was located outboard of the helium purge membrane and fiberglass mat substrate (first two systems in Fig. 7). The gaseous N_2 purged MLI was to provide the lower ground-hold heating rates desired. Difficulty was experienced in maintaining a sealed purge membrane and accurately controlling the thickness of the helium-purged substrate (localized compression). This caused some N_2 condensation within the MLI (or elsewhere on cold surfaces within the vacuum chamber) which significantly increased the pumpdown times to achieve on-orbit performance levels (not achieved in over 500 min of testing). This concept holds considerable promise if the proper balance between the thicknesses of the helium purged sublayer and the nitrogen purged MLI is achieved since the MLI is directly exposed to the external environment and will rapidly evacuate during ascent.

Finally, MSFC currently has an in-house test program underway investigating the thermal performance of a foam-MLI insulation system on a 34.5 ft^3 tank.[13] The insulation system consists of 3/8 in. of SOFI (spray-on foam insulation) and 15 layers of MLI. The system was designed for relatively short missions (e.g., propellant resupply), but will provide important data on a foam/MLI system during a ground-hold condition, transient performance during rapid pumpdown, and on-orbit performance for a range of hot-side boundary temperatures.

CANDIDATE WET-LAUNCHED MLI SYSTEMS

As mentioned previously, a wet-launched LH_2 tank will require some means of protecting the MLI from condensation or freezing of gases within the MLI system during ground hold and ascent operations. This can be accomplished by the use of a sealed sublayer of foam[13,14] or a helium purge or combinations of helium and nitrogen purged MLI. One of the criteria used in selecting a particular system is the magnitude of the ground-hold heating rate because the effective density of the contained fluid (and hence, tank size) is influenced by the heat input. Higher heating rates result in lower effective densities, which means larger tanks for a given propellant requirement. Figure 8 compares the calculated ground-hold heating rates as a function of MLI thickness for several MLI system concepts. The data assumes natural convection heating from both a 500 °R (40 °F) and from a 540 °R (80 °F) ambient temperature environment. In reality, some applications may have some forced convection heating, but the curves are adequate for comparing the various insulation concepts.

The SOFI-MLI combination gives the lowest heating rates and the helium-purged system the highest. Also shown is an estimate of a helium purged system using the Shuttle/Centaur purge bag concept[6] where the two outboard vented shields provided a total gap of about 1/8 in. of dry N_2 space. As mentioned previously, the performance of the system with 1.5 in. of helium purged foam (instead of the MLI on the figure) was about 100 Btu/hr ft^2 which is close to the prediction. The remaining curve for the combined helium/nitrogen purged system gives heating rates between the extremes and which is biased toward the lower heating rates.

The insulation assumed for the foam-MLI combination is SOFI (sprayed-on foam insulation) that is currently under test at MSFC.[13] Spotted on the curve (circle symbols) is the SOFI thickness required to maintain a 250 °R temperature at the foam surface. The SOFI thickness required is approximately 70 to 80 percent of the MLI thickness for the 250 °R temperature assumed. This temperature could be lowered to achieve thinner foam thicknesses, but presents a risk because inadvertent fluffing of the insulation can significantly drop the foam surface temperature and may cause condensation or freezing of N_2 beneath the MLI. This can also occur if cracks are formed in the foam and once it occurs the MLI performance can be seriously degraded during pumpdown in orbit. It is seen that relatively thick layers of foam will be required for the thicker MLI systems needed for the lunar missions. For example, 1.5 in. of MLI will require about a 1-in. layer of SOFI even for the higher ambient temperature, giving a total thickness of 2.5 in. This foam then will also add some heat capacitance that will be absorbed by the LH_2 tank once on-orbit conditions are obtained.

The combined helium/nitrogen purged MLI assumes that the membrane separating the two purge gases be maintained at 250 °R. This system also has the same risks as the SOFI-MLI when fluffing occurs. This concept has the advantage of relatively good performance during ground-hold and will approach space-hold conditions more rapidly than a pure helium purged system, because there is less MLI beneath the purge bag membrane to evacuate. For an ambient temperature of 540 °R and total thickness of 1.5 in., the purge membrane for the combined purge system should be located 0.36 in. from the outer surface (location of the purge membrane will decrease to 0.3 in. for an ambient temperature of 500 °R). Again, this concept will require strict control of the relative thicknesses of the helium and nitrogen purged MLI.

The helium purged systems give the highest ground-hold heating rates, but provide a positive means of insuring no condensate forms within the MLI. Additional valves and purge lines will be required and an effective method of venting the purge bag during ascent will also be needed (this was developed for Shuttle/Centaur as reported in Ref. 6). One potential advantage that arises from the relatively high ground-hold heating rates is that the whole insulation system is sub-cooled and has surface temperatures more closely approaching that which will be experienced once on-orbit (i.e., 400 to 450 °R). More importantly, advantage may be taken of the nearly linear temperature profile imposed across the MLI during the ground-hold helium purge. Once the insulation system is evacuated, the MLI temperature will adjust to its steady state temperature profile dictated by the on-orbit radiant heating environment. In order to achieve this, a good portion of the incoming heat must go into warming the various layers from the outside in. The colder inner portions of the MLI will also have lower conductivities through the spacers and lower emittances due to the lower temperatures which should further reduce the heat reaching the tank during the transient. This is shown conceptually in Fig. 9 where the insulation performance and temperature profiles are qualitatively estimated for a $1\frac{1}{2}$ in. thick layer of helium purged MLI for ground-hold, transient pumpdown, and average on-orbit steady state conditions. Again, a Shuttle/Centaur type purge bag with two vented outboard shields are used. The advantage of this system is that the vented shields will evacuate quickly and give a relatively low heat transfer rate (<1 Btu/hr ft^2) while the remainder of the MLI beneath the purge bag evacuates. The number of vented outboard shields could also be increased if needed.

Depending on how long the insulation remains in its transient prechilled state and how rapidly the MLI is evacuated, a significant performance benefit may be achieved. Thicker layers of MLI should provide longer times yet for the insulation to achieve its on-orbit steady state performance. It should be noted that this system depends on effectively venting the purge bag during ascent in order to achieve an expedient pumpdown of the MLI. It is expected that this pumpdown of the MLI within the purge bag will take longer than when the MLI is exposed directly to the vacuum environment of space as was the case for the systems shown in Fig. 7.

Carrying the idea of MLI subcooling further, additional helium purged foam or fiberglass mat could be applied outboard of the MLI but still within the purge containment membrane, or layers of foam and/or additional MLI could be placed outboard of the purge containment membrane. These concepts are depicted conceptually in Fig. 10 for a nominal 1.5 in. of MLI. All of these concepts will significantly prechill the purged MLI during ground-hold operations, hence leading to long periods on-orbit where the MLI could have an improved performance over that expected at steady state conditions. An additional benefit is also derived from the reduced ground-hold heating rates.

It is seen that all three of these methods will force the bulk of the MLI to temperatures less than 250 °R during ground-hold operations.

In the first concept shown in Fig. 10, a 1.5 in. layer of fiberglass mat or a low-density foam is placed between the MLI and the outer purge bag system. Again, a Shuttle/Centaur (S/C) type purge bag with outboard vented shields is used to quickly achieve a reasonably low heat transfer rate once on-orbit. In this application the foam or mat could be bonded in small sections to the inside of the helium purge membrane. The purge membranes tend to inflate during ground purge, so a minimum of 3 in. of dead helium space would be provided (helium purge rate only replaces natural leakage from the system, hence the flow rates are minimized). The ground-hold heat transfer for this system will be equivalent to that for a 3 in. thickness of the S/C system shown in Fig. 8, that is, 60 to 70 Btu/hr ft^2, and all of the MLI will be preconditioned to temperatures less than 250 °R.

In the combined N_2 purge/He purged concept shown, approximately 1.2 in. of MLI will be at temperatures of 250 °R or less. The remainder of the MLI outboard of the purge bag is directly vented to quickly provide lower heating rates once on-orbit. In this concept it is important to avoid local compressing of the MLI beneath the purge bag or local fluffing of the MLI outboard of the purge bag. This can be achieved by providing positive density control of the MLI as was done in Ref. 7. The estimated ground-hold heating rate for this system is between 70 to 74 Btu/hr ft^2.

In the last concept on Fig. 10, a thin layer of foam could be bonded (attached) to the outboard surface of the purge bag beneath the vented MLI to provide a positive means of controlling the MLI surface temperature at 250 °R. The advantage of the foam is that its thickness does not change and hence provides a fixed thermal resistance. The only remaining task then is to avoid local compression of the MLI. The foam layer has not been sized and is only conceptually shown. The ground-hold heat transfer rates would be expected to fall between the other two concepts discussed on Fig. 10.

All of the purged concepts discussed have higher heating rates than the foam (SOFI)/MLI systems, but may perform better overall due to their more positive methods of avoiding condensibles or freezing of gases in the MLI and due to possible gains resulting from insulation system subcooling during ground operations. Because of the wide range of MLI applications for the SEI missions, both the purged MLI systems and the foam-MLI should be aggressively pursued in future ground testing programs so that a data base can be obtained to optimize the thermal protection system for each specific mission scenario of interest.

CONCLUDING REMARKS

In reviewing the MLI requirements for the SEI missions, it is seen that generally thick MLI systems (>1.5 to several inches) will be required for LH_2 tankage and that the hot side boundary temperatures will range from <100 °R to >600 °R. This will require an extension of the existing data base for MLI which currently has considerable data at thicknesses less than 1.5 in. with hot side boundary temperatures primarily at 530 °R. Also, there is a wide difference between calorimeter data and tank-applied data due to seams and penetrations which implies that an enhanced technology program in this area could have potentially high payoffs. This is especially true for the thicker systems where the seam and penetration effects are expected to cause still higher degradations in MLI performance.

There are several methods of applying MLI to wet-launched LH_2 tanks in order to preclude the condensing or freezing of gases within the insulation system. These generally include the foam/MLI systems or the helium purged systems. The foam/MLI systems give excellent ground-hold performance, but the foam must be flaw-free and must be carefully designed to insure that cryopumping or condensation and freezing do not take place underneath the MLI. Also, for thick MLI, relatively thick foam layers will be required (~70 to 80 percent of MLI thickness), and the heat contained within the foam will be eventually absorbed by the tank once on-orbit.

The helium purged systems represent a positive means of preventing condensation or freezing within the MLI. However, they generally have higher ground-hold heating rates, will require additional system weights to purge and vent the MLI, and will take longer to evacuate any MLI beneath the purge bag. One advantage of the helium purged systems is that it prechills the MLI system during ground-hold operations and limited test data indicates this may have a beneficial effect on the total heat absorbed by the system in the transient post-launch performance of the system. Methods of further prechilling the MLI during ground-hold (and hence decreasing the ground-hold heat transfer rate) include adding foam or fiberglass between the purge bag and the MLI or adding foam and/or vented shields outboard of the purge bag.

In order to optimize any of these systems, both the ground-hold performance as well as the transient performance during evacuation to on-orbit quasi-steady state conditions are required. There is very little data available, either calorimetric or tank applied, on the transient performance of MLI systems and no data, to the author's knowledge, on thick (>1.5 in.) MLI systems applied to LH_2 tank surfaces. Tests should be performed on candidate insulation systems (both foam/MLI and various purged systems) that simulate the pressure environment as well as the thermal environment expected for the wet-launched LH_2 tanks. Integrated heat flux data should be obtained during transient pumpdown tests as well as steady-state tests to determine if the mechanical forces during evacuation of the thick MLI systems cause any degradation in on-orbit performance.

Finally, using the data base generated for thicker MLI systems, optimized tank applied insulation systems should be tested on representative sized LH_2 tanks. The tests should simulate ground-hold, ascent pressure and thermal environment and on-orbit conditions for a sufficient time to reach quasi-steady state conditions for the systems intended mission.

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APPENDIX A

Multilayer Insulation Performance for 530 to 37 °R Boundary Temperatures

The data on the Fig. 1 represent typical MLI performance for boundary temperatures of 530 to 37 °R. Some of the data presented were acquired at slightly different temperatures and have been adjusted to 530 °R by the ratio of the fourth power of the temperatures. The dark symbols are for tank applied data whereas the open symbols represent calorimetric data.

A brief description of each system is given in Table A1 along with the data source. Some of the references from which the data were obtained are not generally available, but are given for completeness.

The "Lockheed Equation" referred to on the figure is from "Thermal Performance of Multilayer Insulations," NASA CR134477, April 1974 (Ref. 5 of this report).

The prediction for "analytical radiation" is from the following equation:

$$\frac{Q}{A} = \frac{\sigma(T_H^4 - T_C^4)}{\frac{1}{\varepsilon_H} + \frac{1}{\varepsilon_C} - 1 + N(\frac{2}{\varepsilon} - 1)}$$

where

Q/A heat transfer rate, Btu/hr ft^2

T_h hot side boundary temperature, °R

T_c cold side boundary temperature, °R

ε_h emittance of hot surface

ε_c emittance of cold surface (usually the tank or calorimeter)

ε assumed emittance of shields (in this case, ε = 0.05)

N number of MLI radiation shields

σ Stefan-Beltzmann constant

<u>REFERENCES FOR MLI PERFORMANCE (FIG. 1) FOR 530 to 37 °R BOUNDARY</u>

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l. Experimental Evaluation of a Purged Substrate Multilayer Insulation System for LH2 Tankage, R.L. DeWitt and M.B. Mellner, NASA TD D-6331, 1971.

TABLE A1

MLI Details for MLI Performance (Figure 1) for 530°R - 37°R Boundary Temperatures

Symbol	Ref	MLI Description	Test Article
□	a	DBL Alum. Mylar with DBL silk net spacers, spiral wrapped	30" dia. calorimeter
△	b	DBL Goldized Mylar with DBL silk net spacer	4' dia. calorimeter
▢	c	DBL Alum. Mylar with Dexiglass or silk net spacers	30" dia. calorimeter
○	d	DBL Alum. Mylar with DBL silk net spacers. Button-pin attachment with Schjeldahl X-850 cover sheets	44" flat plate calorimeter
◠	e	Centaur forward bulkhead insulation. DBL Alum. Mylar with Dimplar spacers	9.5" dia. calorimeter
⊠	f	Schjeldahl X-850 shields with no spacers	9.5" dia. calorimeter
◺	g	Titan/Centaur forward bulkhead radiation shields	9.5" dia. calorimeter
◿	g	Titan/Centaur sidewall radiation shields	9.5" dia. calorimeter
◇	g	Titan/Centaur LO_2-sump shields tested at LH_2 temp.	9.5" dia. calorimeter
▼	h	Superfloc	25" dia. guarded tank
●	i	DBL Alum. Mylar with DBL silk net spacers. Button-pin attachment with Schjeldahl X-850 cover sheets	87" dia. tank
▶	j	Same as above	57" dia. tank
◀	k	22 layers of goldized Superfloc between two face sheets attached with twin-pin fasteners in each blanket. Two blankets	87" dia. tank
■	l	30 layers DBL Alum. Mylar with Dexiglass spacers	82.6" dia. tank

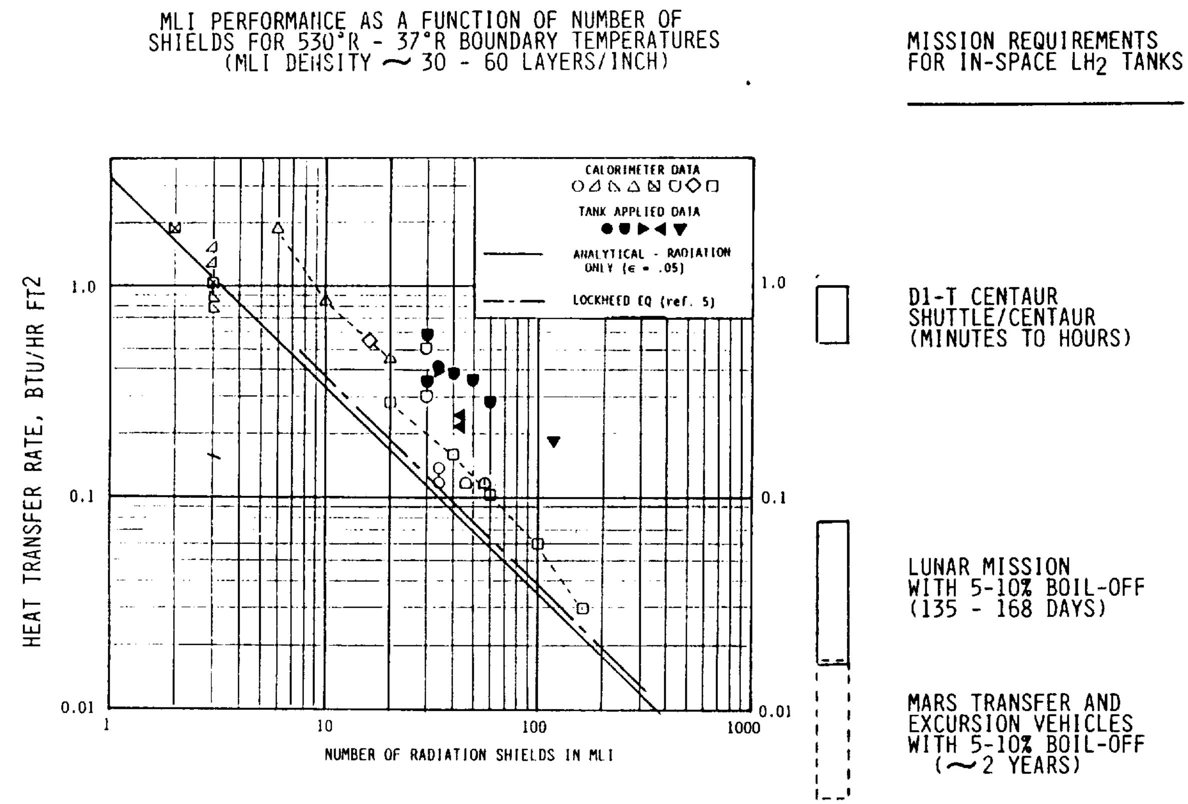

Figure 1. - Multilayer insulation (MLI) preformance vs SEI mission requirements

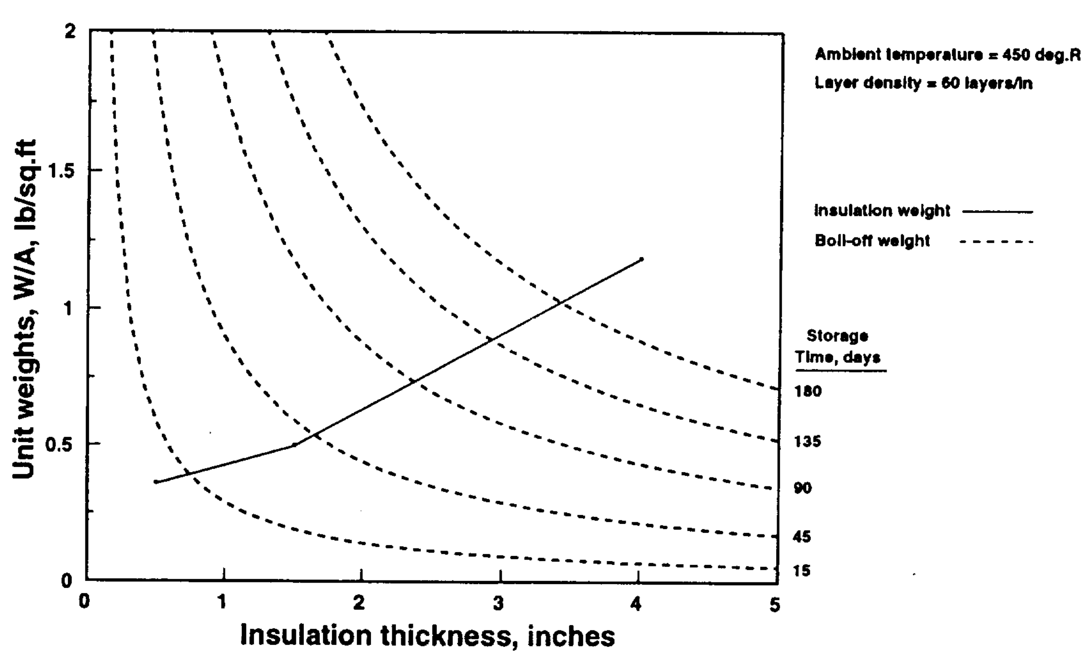

Figure 2. - Typical TPS weights for lunar mission MLI insulated LH2 tanks

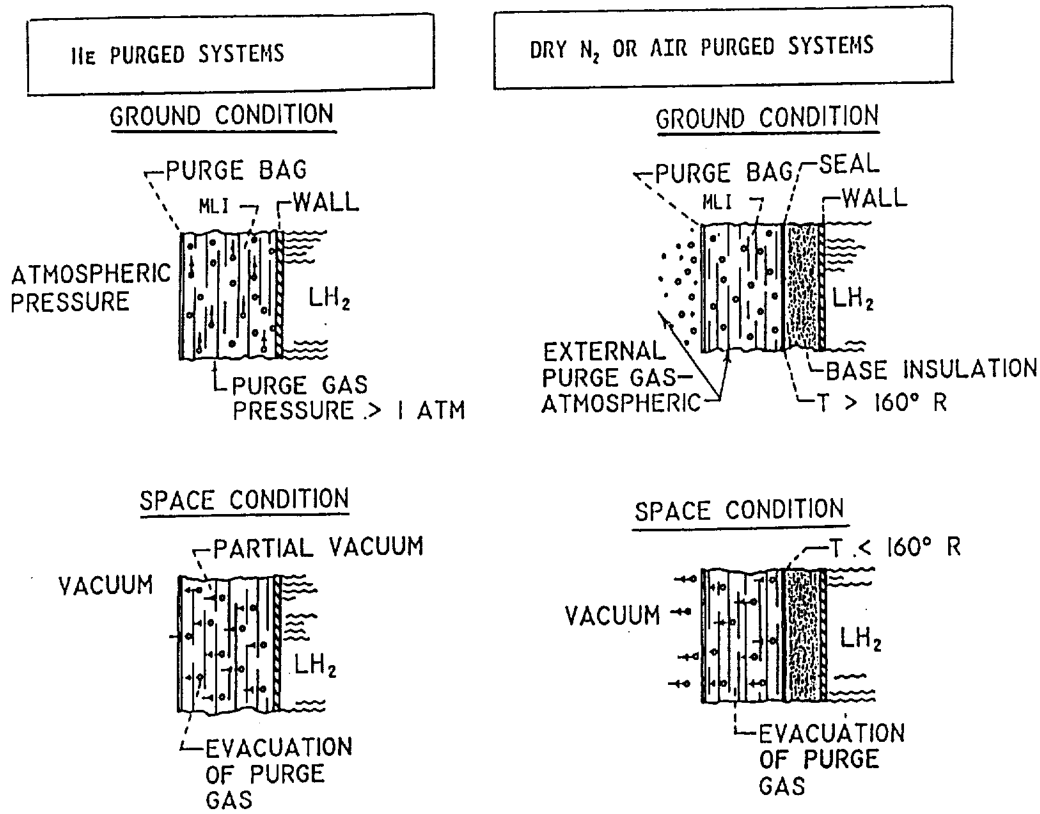

Figure 3. - Lightweight methods for applying MLI to wet-launched LH2 tanks

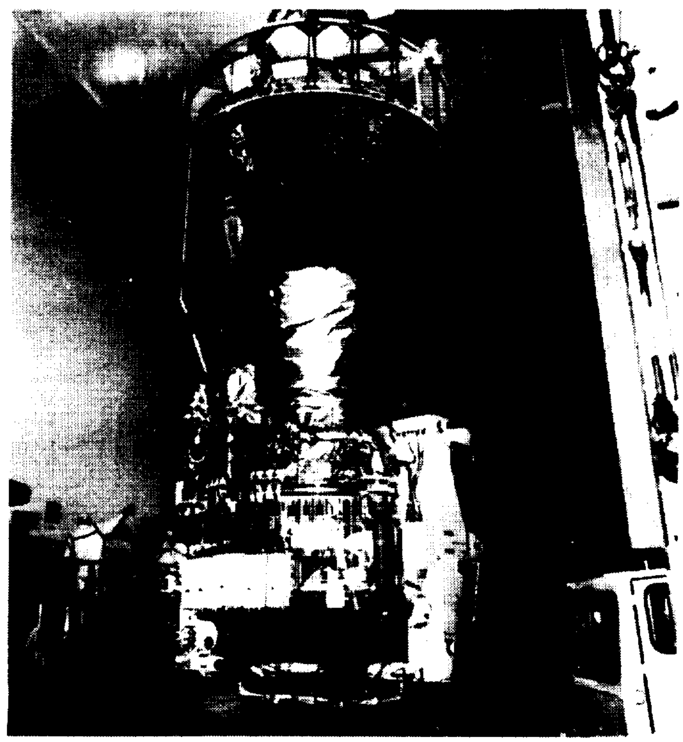

Figure 4. - Centaur vehicles are only flight application of MLI to light-weight cryogenic tankage

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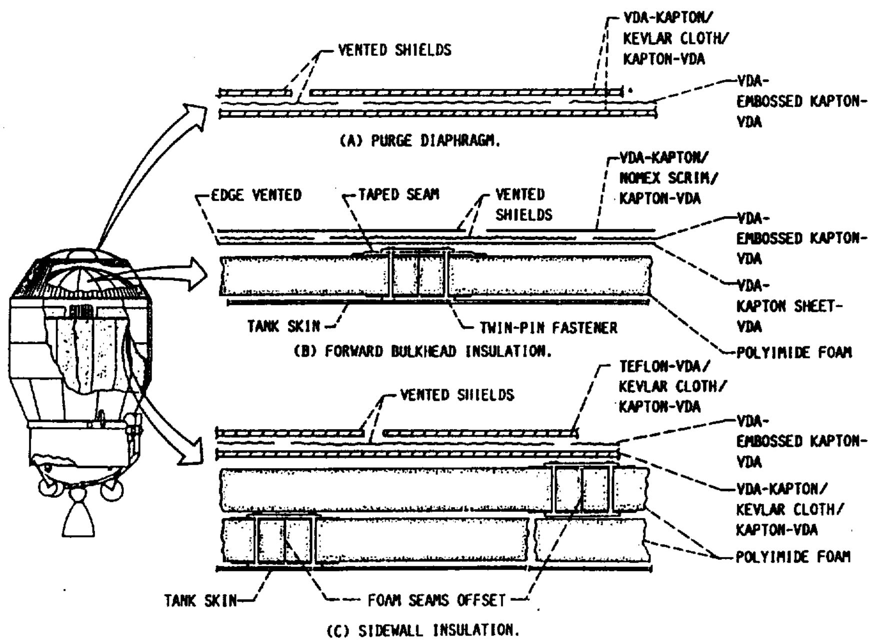


Figure 5. - Primary thermal protection system used on Shuttle/Centaur G-prime LH2 tank

Figure 6. - 87 inch diameter MLI insulated LH2 tank tested at NASA LeRC K-site for both near-earth, lunar surface and deep space conditions

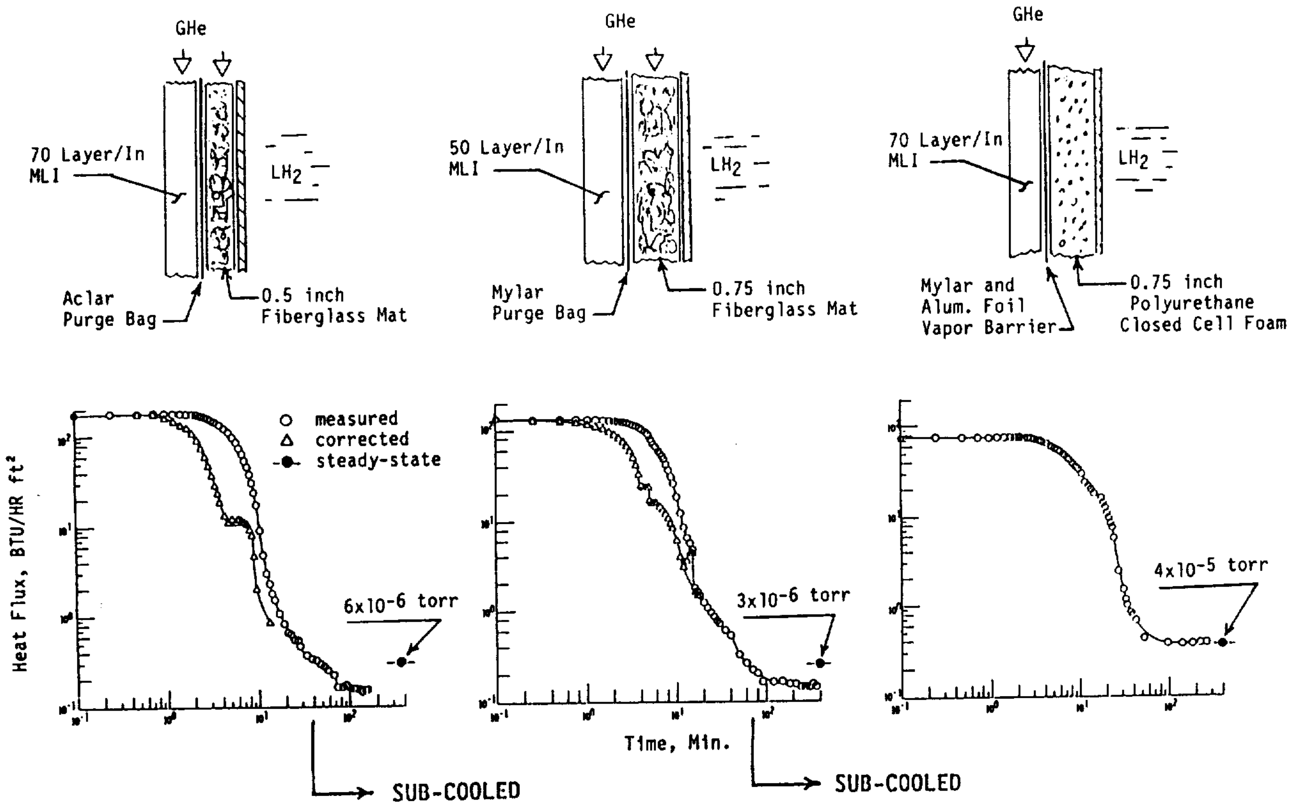


Figure 7. - Transient performance of 30 layer, helium-purged MLI systems during simulated ascent (Ref. 4)

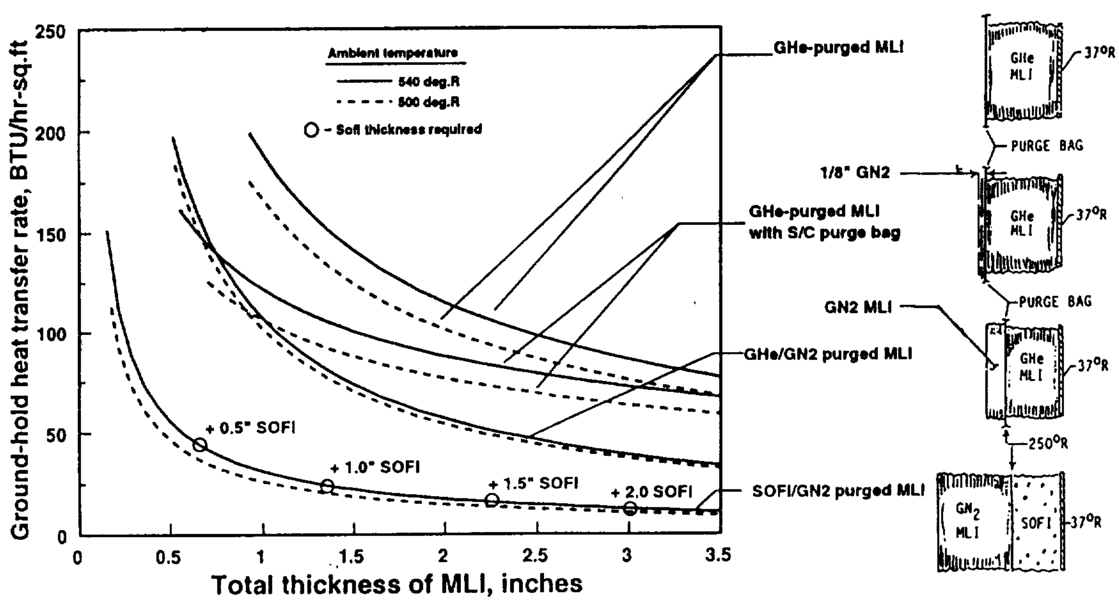
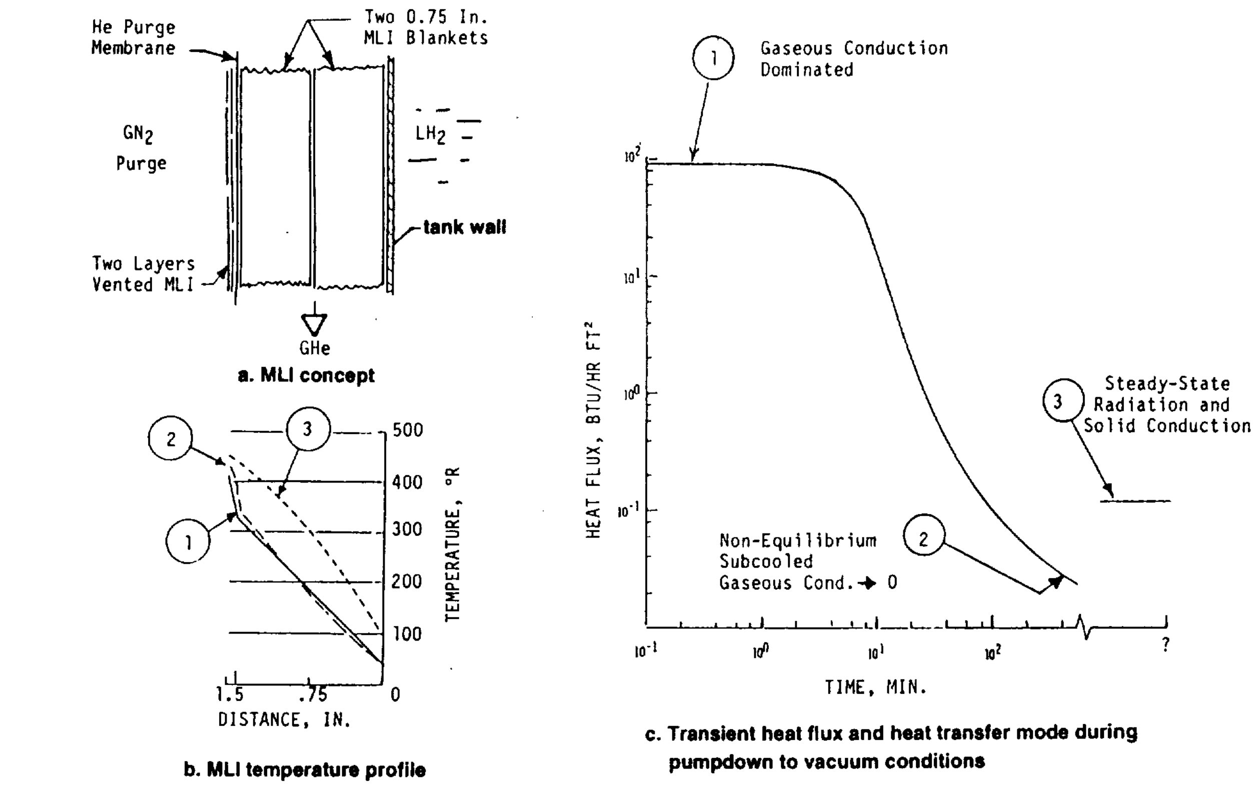


Figure 8. - Comparison of ground hold heat rates for typical wet-launched MLI systems for LH2 tanks



a. MLI concept

b. MLI temperature profile

c. Transient heat flux and heat transfer mode during pumpdown to vacuum conditions

Figure 9. - Estimated transient performance of helium-purged MLI for a wet-launched LH2 tank (ambient temperature = 500 degree R.)

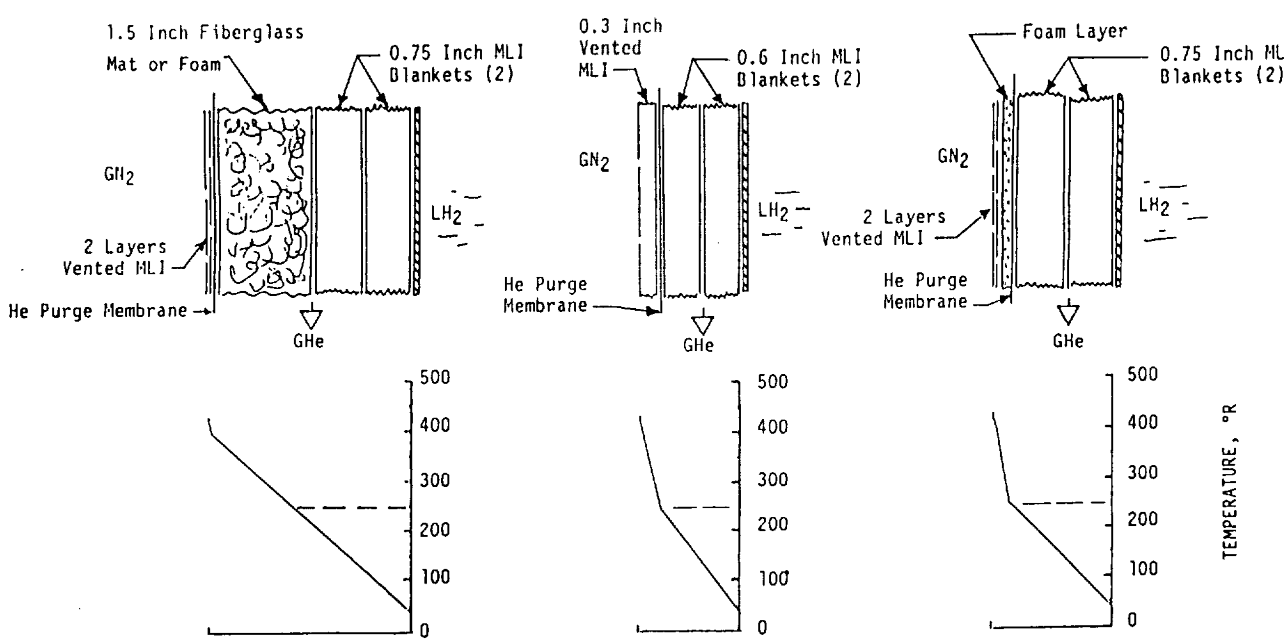


Figure 10. - Concepts for sub-cooling MLI, during ground-hold, for wet-launched tanks (ambient temperature = 500 degree R.)

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Report Documentation Page

1. Report No. NASA TM-104493 AIAA-91-2176	2. Government Accession No.	3. Recipient's Catalog No.

4. Title and Subtitle

A Review of Candidate Multilayer Insulation Systems for Potential Use on Wet-Launched LH_2 Tankage for the Space Exploration Initiative Lunar Missions

5. Report Date

6. Performing Organization Code

7. Author(s)

Richard H. Knoll, Robert J. Stochl, and Rafael Sanabria

8. Performing Organization Report No.

E-6351

10. Work Unit No.

506-48

9. Performing Organization Name and Address

National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135-3191

11. Contract or Grant No.

13. Type of Report and Period Covered

Technical Memorandum

12. Sponsoring Agency Name and Address

National Aeronautics and Space Administration
Washington, D.C. 20546-0001

14. Sponsoring Agency Code

15. Supplementary Notes

Prepared for the 27th Joint Propulsion Conference cosponsored by the AIAA, SAE, ASME, and ASEE, Sacramento, California, June 24-27, 1991. Responsible person, Richard H. Knoll, (216) 433-2419.

16. Abstract

The storage of cryogenic propellants such as liquid hydrogen (LH_2) and liquid oxygen (LO_2) for the future Space Exploration Initiative (SEI) will require lightweight, high performance thermal protection systems (TPS's). For the near-term lunar missions, the major weight element for most of the TPS's will by multilayer insulation (MLI) and/or the special structures/systems required to accommodate the MLI. Methods of applying MLI to LH_2 tankage to avoid condensation or freezing of condensible gases such as nitrogen or oxygen while in the atmosphere are discussed. Because relatively thick layers of MLI will be required for storage times of a month or more, the transient performance from ground-hold to space-hold of the systems will become important in optimizing the TPS's for many of the missions. The ground-hold performance of several candidate systems are given as well as a qualitative assessment of the transient performance effects.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Propellant storage Cryogenics Thermal control	Unclassified - Unlimited Subject Categories 28 and 15

19. Security Classif. (of the report)	20. Security Classif. (of this page)	21. No. of pages	22. Price*
Unclassified	Unclassified	20	A03

NASA FORM 1626 OCT 86

*For sale by the National Technical Information Service, Springfield, Virginia 22161